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PROGRESSIVE DAMAGE MODELING IN COMPOSITE SHELL STRUCTURES

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The objective of this project was to develop a computational model for simulating progressive damage growth around a notch in a composite shell structure under general membrane and flexural loads. This was accomplished using nonlocal continuum damage mechanics wherein damaged material was treated as exhibiting a strain-softening response. This report is a brief summary of the results of this project. More detailed descriptions can be found in References 1-3.

Damage tolerance requirements play a key role in the design of composite shell structures in the commercial aircraft industry. Most of the theories that have been proposed for predicting failure of composite laminates with cracks contain empirical parameters that have validity for only a limited range of crack sizes. A complicating factor is the development of a damage zone of considerable influence in the vicinity of the crack tip. Recent research indicates that the damage manifests itself in the form of strain-softening of the laminate material, i.e., a stress-strain response that exhibits a decrease in stress with increasing strain beyond some critical strain. Modeling strain-softening by means of classical plasticity theory results in spurious mesh sensitivity in a finite element analysis due to the tendency of the damage zone to localize to a zero volume. An effective technique for overcoming this difficulty is to treat the material as having some of the properties of a nonlocal continuum, i.e., a body where certain quantities are functions of strain averaged over an area of the body. Using ideas from nonlocal elasticity theory and continuum damage mechanics, we have formulated two approaches for meeting the objective stated above.

In the first approach the analysis of a composite laminate was not done on a ply-by-ply basis, but rather it was simply considered as a homogeneous material with orthotropic membrane and flexural stiffness. Using an elastic energy equivalence concept, the stiffness relations for damaged material were determined. For this simple model we assumed that when damage occurs, it is solely a function of the middle surface nonlocal strains which are weighted averages of the strains over an area around a point. The principal damage directions were assumed to coincide with the principal material directions, and both membrane and flexural stiffness were assumed to be equally affected by damage development. The damage variables contain empirical parameters that are functions of the strain-softening properties of the laminate and a characteristic length that influences the weight functions used to compute nonlocal strains. This model was incorporated into a finite element formulation that made use of a quadrilateral shell element. The first case analyzed was a center-cracked laminated plate under simple tension. Two laminates were considered: one exhibiting brittle strain-softening response and the other ductile strain-softening response. Calculations were compared to test results, and the theory was shown to accurately predict failure in these two laminates over a wide range of crack sizes. Next, an analysis was performed on a section of a composite aircraft fuselage containing a crack and subjected to internal pressure loading. Again calculations were compared to test results. The failure pressure predicted by the model was within one percent of that experienced in the test. A comparison of calculated and measured strains at several locations ahead of the crack tip indicated good agreement during the elastic portion of the response. However, the experimental pressure versus strain curves exhibited strongly nonlinear behavior at a pressure about 10 percent below that of the theoretical curves. Due to the complexity of the response of the panel in the test, it is not clear whether this discrepancy was caused by inaccuracy in the softening model or by frame-skin debonding that occurred during the test but was not modeled in the analysis.

The second approach assumed that a simple linear variation of damage through the shell thickness could adequately represent what is likely an erratic phenomenon. This assumption enabled the damage and stiffness of each ply to be represented by bulk laminate stiffness properties. The linear variation of damage is represented by $D(z) = D^0 + \alpha z$. Parameter D^0 is the same damage parameter used in the first approach, and its effect upon bulk laminate stiffness was the same except in cases of shear. However, the introduction of parameter α represents asymmetry of damage through the thickness of the laminate, and it was shown to couple laminate extension and bending just as would exist if a laminate were composed of nonsymmetric ply stacking. D^0 was again made dependent on nonlocal mid-plane strain, and α was made dependent on curvature as well as a compliance term which represented how easily surface plies develop damage. A nonlinear tension plus flexure test was developed to characterize the compliance term for the ATCAS crown panel skin. The test revealed fracture to be sensitive to flexure. However, finite element analysis, that incorporated the damage model, over-predicted the fracture strength; and the use of various compliance control methods indicated that α has a minor effect on strength. This suggests that D^0 should be dependent on curvature as well as mid-plane strain according to some interaction criterion. Indeed, the developed theory indicated this to be the case, but it was ignored in order to try the simple model first. Next, analysis was performed on the same cracked fuselage discussed above. In this case the analysis under predicted failure by 10%; however, surface strains measured in the crack path showed good agreement into the nonlinear regime. Both analysis and test indicated that flexure was mild in the area of the crack and probably had little effect on fracture.

Both approaches employed the same bulk damage growth function, and predictions of tension fracture strength were in agreement. However, the second approach also allows for damage growth under shear conditions, such as exists on the fuselage side quadrant. Once a curvature and mid-plane strain interaction criterion is developed for damage parameter D^0, high flexure regions on aircraft structure can be analyzed for damage tolerance, such as exist in the skin adjacent to substructure. Therefore, this analysis represents a promising tool for assessing damage tolerance of complex laminate structure.

REFERENCES

1. Kennedy, T. C. and Nahan, M. F., "A simple nonlocal damage model for predicting failure of notched laminates," *Composites Structures*, in press.

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3. Nahan, M. F. and Kennedy, T. C., "Laminated plate damage mechanics and nonlocal damage evolution," submitted to *Composites Science and Technology.*